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RESEARCH MEMORANDUM

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LOW-SPEED CHARACTERISTICS IN PITCH OF A 34° SWEPT FORWARD
WING WITH CIRCULAR-ARC AIRFOIL SECTIONS

By

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LOW-SPEED CHARACTERISTICS IN PITCH OF A 34° SWEEPFORWARD

WING WITH CIRCULAR-ARC AIRFOIL SECTIONS

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SUMMARY

An investigation was conducted in the Langley 19-foot pressure tunnel to determine the lift, drag, and pitching-moment characteristics of a 34° sweptforward wing tested alone and with various combinations of extensible round-nose leading-edge flaps, trailing-edge split flaps, and a representative fuselage. The wing had circular-arc airfoil sections, an aspect ratio of 3.9, and a taper ratio of 0.625. The fuselage had a fineness ratio of 12 and was tested in low-wing, midwing, and high-wing combinations. Test Reynolds numbers ranged from 3,100,000 to 9,600,000.

The maximum lift coefficient of the basic wing was 0.78 and unstable changes in pitching moment occurred just below maximum lift. The addition of leading-edge flaps to all fuselage-off configurations produced substantial increases in maximum lift coefficient but caused undesirable variations in pitching moment below the stall. The addition of the fuselage to the plain wing produced a destabilizing effect by increasing the longitudinal-stability parameter dC_m/dC_L by 0.16 in the low-lift range, reduced the undesirable variations in pitching moment in the high-lift range, and increased the maximum lift coefficient by an amount which varied from 0.14 to 0.19, depending on fuselage position. The addition of full-span leading-edge flaps to the midwing-fuselage combination increased the maximum lift coefficient to 1.30 without seriously decreasing the longitudinal stability. With this combination and for an assumed sinking speed of 25 feet per second, the glide speed would be 112 miles per hour for a wing loading of 30 pounds per square foot. For all combinations, half-span split flaps did not appear promising as a means of increasing lift because of large changes in trim and large unstable deviations in longitudinal stability near the stall. A variation of Reynolds number from 3,100,000 to 9,600,000 had no appreciable effect on the aerodynamic characteristics of the wing.

INTRODUCTION

It has been demonstrated that wing sweep moderates the large, undesirable aerodynamic changes encountered at transonic speeds and reduces wing drag at high speeds. Theory indicates that the shock drag at supersonic speeds is considerably less for sharp-nose airfoils than for conventional sections. Thus, for operations at supersonic speeds, an advantage is obtained by the use of sweep and sharp leading edge. Previous investigations have indicated, however, that both wing sweep and sharp-nose airfoils can engender serious detrimental effects with respect to the low-speed longitudinal stability, the attainable maximum lift, and the stalling characteristics of a wing. It was considered desirable, therefore, to investigate the low-speed characteristics of a swept wing having sharp-nose profiles. Tests were made in the Langley 19-foot pressure tunnel of such a wing in both sweptforward and sweptback arrangements. Tests of this wing sweptback 42° are reported in reference 1. The aerodynamic characteristics in pitch of the wing having the leading edge sweptforward 34° are presented herein. The wing has circular-arc airfoil sections, an aspect ratio of 3.9, and taper ratio of 0.625.

Tests of the plain wing were made alone and in combination with various high-lift and stall-control devices including half-span split flaps, extended round-nose leading-edge flaps, and upper surface flaps. A representative fuselage was tested in high-wing, midwing, and low-wing positions. Most of the tests were conducted at Reynolds number values of 3,100,000 and 6,900,000.

COEFFICIENTS AND SYMBOLS

The data are referred to the wind axes. Pitching moments are referred to the quarter-chord point of the mean aerodynamic chord located in the wing-chord plane as indicated in figure 1. The dimensions and area of the basic wing are used in reducing all data to coefficient form.

C_L	lift coefficient (L/qS)
C_D	drag coefficient (D/qS)
C_m	pitching-moment coefficient (M/qSc)
R	Reynolds number ($\rho Vc/\mu$)
M_o	Mach number (V/a)

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α	angle of attack of wing-chord plane
dC_m/dC_L	rate of change of pitching-moment coefficient with lift coefficient
where	
L	lift
D	drag
M	pitching moment
S	wing area
$\bar{c}$	mean aerodynamic chord measured parallel to plane of symmetry $\left(\frac{2}{S} \int_0^{b/2} c^2 dy \right)$
$\bar{x}$	distance from leading edge of root chord at plane of symmetry to quarter-chord point of mean aerodynamic chord $\left(\frac{2}{S} \int_0^{b/2} cx dy \right)$
x	longitudinal distance, parallel to plane of symmetry, from leading edge of root chord to quarter-chord point of each section
c	local chord measured parallel to plane of symmetry
b	wing span
y	spanwise coordinate
q	free-stream dynamic pressure $\left(\frac{1}{2} \rho V^2 \right)$
V	free-stream velocity
ρ	mass density of air, slugs per cubic foot
μ	coefficient of viscosity
a	velocity of sound

MODEL

The plan and elevation of the wing and fuselage are shown in figure 1. The wing has an aspect ratio of 3.94 and a taper ratio of 0.625. Each wing panel was fabricated from a solid steel blank by making both the upper surface and the lower surface a section of a cylinder with a radius of 83.26 inches. The leading and trailing edges of the wing are sections of the ellipse formed by the intersection of two cylinders, the axes of which intersect at the proper angle to produce the desired taper ratio. The maximum deviation between the leading and trailing edges and the straight lines connecting the leading and trailing edges of the root and tip chords is about 0.4 inch. The leading-edge sweep angle of -33.87° is defined by the straight line connecting the leading edges of the root and tip chords. The line of maximum thickness (fig. 1) lies in the plane containing the axes of the cylinders. In planes perpendicular to this line, the root and tip sections are 10- and 6.4-percent thick, respectively. In a plane parallel to the model plane of symmetry the sections at the root and tip are 7.9- and 5.2-percent thick, respectively. The wing tips start at the $0.975\frac{b}{2}$ station and are rounded both in plan form and in cross section.

The fuselage has a fineness ratio of 12 and circular cross sections, the maximum diameter being 40 percent of the wing root chord (fig. 1). The section of the fuselage intersected by the wing has a constant diameter. The vertical location of the wing with respect to the fuselage axis is defined in terms of the fuselage diameter. Three positions were investigated: 40 percent below, 0 percent, and 40 percent above, thus simulating the low-wing, midwing, and high-wing types of airplanes. No fillets were used at the wing-fuselage juncture. Both the steel wing and the laminated mahogany fuselage were lacquered and sanded to obtain aerodynamically smooth surfaces.

The various flaps are detailed in figure 2 and are shown mounted on the wing in figure 3. The split flaps extend over the inboard 50 percent of the wing span. To simplify the fuselage installation a section of the split flaps (12.3 percent of the wing span) was removed at the wing center for the fuselage-on tests. The flap chord is 20 percent of the wing chord and the flaps are deflected 60° below the wing lower surface as measured in a plane normal to the 80-percent chord (hinge) line. The extended leading-edge flap is a flat sheetmetal plate faired tangentially to a

$\frac{1}{2}$ -inch diameter tube to form a round-nose leading edge. The

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flap has a constant chord amounting to $0.13\bar{c}$ as measured parallel to the plane of symmetry. As measured in a plane perpendicular to the wing leading edge, the upper surface of the flap is deflected 37° below the wing chord. Two spans of the leading-edge flap were tested, one extending from the wing center line to the $0.80\frac{b}{2}$ station and the other extending to the $0.975\frac{b}{2}$ station. The area (unprojected) of the $0.975b$ flap amounts to 13 percent of the wing area. The half-span split flaps were also tested as upper-surface flaps (fig. 3(b)), being deflected -30° from the wing upper surface as measured in a plane normal to the flap hinge line.

TESTS

Tests were made in the Langley 19-foot pressure tunnel with the wing mounted on a two-support system as shown in figure 4. Lift, drag, and pitching moment were obtained at zero angle of yaw through an angle-of-attack range for values of Reynolds number and Mach number as follows:

R	M_o
3,100,000	0.07
5,300,000	.12
6,900,000	.15
9,600,000	.22

Test results of the plain wing combined with the various high-lift flaps are given for $R = 3,100,000$, $R = 6,900,000$, and also for the split-flap configuration, $R = 9,600,000$. The data presented for the fuselage-on combinations were obtained from tests at $R = 6,900,000$ except for the configuration with upper-surface flaps which was limited to $R = 5,300,000$ because of the high stresses in the support system.

Stall characteristics were studied by means of tufts attached to the upper surface of the wing behind the 20-percent chord line.

CORRECTIONS TO DATA

The data are corrected for the effects of jet boundary, air-stream misalignment and tare and interference effects of the support system.

The jet-boundary corrections to the angle of attack and drag coefficient calculated from the jet-boundary induced vertical velocities given in reference 2 are as follows:

$$\Delta\alpha = 0.977C_L$$

$$\Delta C_D = 0.0147C_L^2$$

The correction to the pitching-moment coefficient due to the tunnel-induced distortion of the loading is:

$$\Delta C_m = 0.003C_L$$

The dynamic pressure used in determining the aerodynamic coefficients was corrected for wake blockage, a function of model profile drag, by the following equation:

$$q = q_{\text{uncorrected}} \left[1 + 0.062 (C_D - C_{D_i}) \right]$$

where the induced drag coefficient C_{D_i} was approximated by the equations:

$$C_{D_i} = 0.0796C_L^2 \quad (\text{split flaps off})$$

$$C_{D_i} = 0.0796C_L^2 - 0.0049 \quad (\text{split flaps on})$$

RESULTS AND DISCUSSION

The stall characteristics of the plain wing alone and in combination with leading-edge and trailing-edge flaps are shown in figure 5. The lift, drag, and pitching-moment data for these configurations are presented in figures 6 and 7. The results of the wing-fuselage investigation are presented in figures 8 to 11. Figure 12 presents the longitudinal stability parameter dc_m/dc_L as a function of lift coefficient for various configurations of figure 11. Using a constant sinking-speed value of 25 feet per second, the relation of the glide speed to the wing loading was calculated from the lift-drag variations for two of the configurations of figure 11 and is presented in figure 13.

together with the previously discussed large trim change accompanying split-flap deflection preclude the application of such split flaps as a high-lift device for this wing. The large change in pitching moment accompanying split-flap deflection for all values of lift coefficient suggests the possibility of using split flaps as a pitch-control device.

In order to determine pitch-control effectiveness for a negative flap deflection, the midwing-fuselage combination with leading-edge flaps was tested with the split flaps mounted on the wing upper surface. The displacement in pitching-moment coefficient of 0.10 at zero lift (fig. 11(a)) decreased with increasing lift coefficient and was insignificant above a lift coefficient of 1.10. The decreased effectiveness was associated with the flap operating in a region where the flow gradually separated as the lift coefficient increased.

With split flaps off, the longitudinal stability characteristics of the fuselage leading-edge flap combination were almost as good as those previously noted for the fuselage combinations without any flaps. This may be seen in figure 12(a) where the values of dC_m/dC_L below the stall ranged from 0.05 to 0.25 with leading-edge flaps on and from 0.08 to 0.16 with leading-edge flaps off. At the stall the values became decidedly negative for both conditions. Extending full-span leading-edge flaps on this midwing fuselage combination appears to be an attractive means of increasing the maximum lift coefficient from 0.97 to 1.30 (fig. 11(a)) without seriously decreasing the longitudinal stability.

The drag characteristics of the various combinations shown in figure 11(b) indicate moderate drag values in the low-lift range. The flight study of reference 4 indicated a maximum safe vertical velocity for the landing approach to be 25 feet per second. Using this value of sinking speed, the variation of glide speed with wing loading is given in figure 13 for the split-flap-off configurations of figure 11. With leading-edge flaps extended on the midwing-fuselage combination, the glide speed would be 112 miles per hour for a wing loading of 30 pounds per square foot at an attitude of 16° angle of attack. This would be about 120 percent of the minimum gliding speed as shown by the lift-drag polar of figure 11(b) on which has been superimposed a grid of sinking speed and indicated glide speed for a wing loading of 30 pounds per square foot.

Full-span leading-edge flaps appear to be a satisfactory high-lift device for this swept-forward wing-fuselage combination from the following considerations: (a) extending the leading-edge flaps would cause little change in trim; (b) there would be a fair degree

Fuselage-On Combinations

Split flaps off and on.— The stall diagrams in figures 8(a) and 8(b), presented for the midwing combination only, are also representative of the flow patterns observed for the low-wing and the high-wing fuselage combinations. The region of initial stall on the plain wing was blanketed by the fuselage and the stall progression on the exposed area was about the same as that observed on the wing alone with split flaps either off or on. The results of figures 9 and 10 indicate that the fuselage increased the maximum lift coefficient by increments varying from 0.14 to 0.19, the increments being about in proportion with the amount of wing upper-surface area enveloped by the fuselage in each of the three combinations.

The addition of the fuselage produced a destabilizing effect, amounting to an increase in dc_m/dc_L of 0.16 at zero lift for the flaps-off condition. This large destabilizing effect of the fuselage was a result of the forward location of the fuselage with respect to the wing, an arrangement which is typical for designs incorporating sweptforward wings. No undesirable pitching-moment changes occurred in the high-lift region with the fuselage combined with the unflapped wing (fig. 9(a)). Deflecting split flaps with fuselage on, however, caused large unstable changes in center of pressure (fig. 10(a)). With fuselage off, removing the center section of the split flaps shifted the pitching-moment coefficient about 0.05 throughout the lift range. (Compare figs. 6(a) and 10(a).) As the wing arrangement was changed from the low-wing to the high-wing position, the center gap of the flaps was considerably reduced, and this probably caused the large changes in pitching moment attributed to fuselage position observed especially near the maximum lift.

Midwing fuselage with leading-edge flaps.— As shown in figures 8(c) and 8(d), the stall progression of the wing with $0.975 \frac{b}{2}$ -span leading-edge flaps and midwing fuselage was gradual as contrasted to the stall without a fuselage. This stretching out of the stall progression with the leading-edge flap configurations not only brought about improved longitudinal stability characteristics by decreasing the abrupt undesirable changes in pitching moment (fig. 11(a)) but also eliminated the premature flattening of the lift curve previously noted with split flaps on and fuselage off (fig. 7(b)).

With split flaps on, the longitudinal-stability parameter dc_m/dc_L ranged from below zero to values in excess of 0.40, the upper limit of the graph (fig. 12(b)). This wide range of dc_m/dc_L

The variations of drag and pitching-moment coefficient with lift coefficient and the effect of Reynolds number were consistent with those of the flap neutral configuration.

Leading-edge flaps.— Root stall still occurred on the wing after leading-edge flaps were installed, but the characteristics of the stall and the stall progression were different. The addition of full-span leading-edge flaps eliminated leading-edge inflow throughout the angle-of-attack range (fig. 5(d)) and only mild inflow existed at the trailing edge at angles of attack below 15° . Then the air flow abruptly separated at the wing root section in a sharply defined region. As the angle of attack was increased, the gradual spanwise spread in the area of separated flow extended over the entire wing chord in contrast with the flow patterns noted for the plain wing, where the area of separated flow fanned out predominately along the wing leading edge. Removing the outer sections of the leading-edge flaps caused cross flow and local areas of stall to occur at the outer end of the flaps (fig. 5(c)). No basic change in the root stall was noticed.

The addition of leading-edge flaps increased the maximum lift coefficient to about 1.20 for either the $0.80\frac{b}{2}$ span or the $0.975\frac{b}{2}$ span flap (fig. 7(a)). Leading-edge flaps effected a destabilizing rotation of the pitching-moment curve in the low-lift range and caused irregular variations in pitching moment when initial separation occurred (figs. 5(c) and 5(d)). The first variation was a negative break in pitching moment which is unexplained since root stall would normally be expected to cause the pitching moment to become more positive on sweptforward wings. When leading-edge flaps were extended, the drag coefficient was increased at low angles of attack, probably because the flaps spoiled the flow over the lower surface of the wing; but above a lift coefficient of 0.3, the drag coefficient was reduced (fig. 7(c)).

Leading-edge flaps and split flaps.— Adding split flaps to the leading-edge-flap combinations caused more severe aerodynamic changes. The stall progressions, though not presented, were similar to those of figures 5(c) and 5(d) with split flaps off. The lift curve flattened at an angle of attack of 16° , remained flat for 4° , and then continued upward (fig. 7(b)). The flattening of the lift curve below maximum lift was more severe than that observed when either flap was tested independently on the wing. The pitching-moment variations associated with initial separation and the flattening of the lift curve were extremely large and unstable.

Wing-Flap Combinations with Fuselage Off

Wing alone.— The stall diagrams of figure 5(a) showed that, as the angle of attack was increased, inflow developed over the wing, especially near the wing leading edge. Any region where the direction of flow had a forward component was interpreted as being a stalled region. Stall was first noticed near the leading edge of the wing root at a low angle of attack (6°). As the angle of attack was increased, the region first spread rearward and then gradually fanned out along the wing leading edge until at maximum lift only the rear portion of the outer half of each wing panel was unstalled.

The slope of the lift curve $dC_L/d\alpha$ from figure 6(a) was only 0.044 near zero lift but increased appreciably with increasing angle of attack until it reached a value of 0.057 at an angle of attack of 8° . The expanding region of separated flow on the wing caused a gradual decrease in the slope at higher angles of attack, resulting in a flat lift-curve peak. The maximum value of lift coefficient was only 0.78 at an angle of attack of 22° .

The pitching-moment coefficient, which was slightly positive up to a lift coefficient of 0.6, became decidedly positive as the stalled area at the wing root section caused the center of pressure of each wing panel to move outboard and therefore ahead. At maximum lift the stalled areas along the leading edge had spread far enough outboard to reverse the direction of the pitching-moment curve. Such large changes in center of pressure are considered undesirable.

The drag coefficient (fig. 6(b)) increased rapidly at moderate values of lift coefficient and became extremely large at maximum lift. The data of figure 6 indicate that the effect of Reynolds number was negligible for the range of Reynolds number of these tests. The characteristics of this wing from the standpoint of the low value of maximum lift and negligible effect of varying Reynolds number were in general agreement with the characteristics noted in two-dimensional tests of biconvex sections (reference 3).

Split flaps.— Adding split flaps did not change the stall progression, though the same stall patterns occurred at slightly lower angles of attack (fig. 5(b)). The lift coefficient was increased about 0.3 throughout the angle-of-attack range, in contrast with the decreasing flap effectiveness with increasing angle of attack noted in the test results of this wing sweptback (reference 1). The large displacement in pitching moment of -0.17 , which accompanied the flap deflection, would cause large changes in trim.

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of longitudinal stability for a center-of-gravity location at (or ahead of) the leading edge of the mean aerodynamic chord; and (c) the glide speed would not be excessive for a moderate wing loading and a safe sinking speed. Also, as for all configurations tested, the ailerons would always operate in unstalled regions at the wing tips.

CONCLUSIONS

The main results and conclusions from an investigation of a 34° sweptforward wing with biconvex sections were as follows:

1. The maximum lift coefficient of the basic wing was 0.78 and unstable changes in pitching moment occurred just below maximum lift.

2. The addition of leading-edge flaps to all fuselage-off configurations produced substantial increases in maximum lift coefficient but caused undesirable variations in pitching moment below the stall.

3. The addition of the fuselage to the basic wing produced a destabilizing effect by increasing the longitudinal-stability parameter dC_m/dC_L by 0.16 in the low-lift range, reduced the undesirable variations in pitching moment in the high-lift range, and increased the maximum lift coefficient by an amount which varied from 0.14 to 0.19 depending on fuselage position.

4. Full-span leading-edge flaps added to the midwing-fuselage combination increased the maximum lift coefficient to 1.30 without seriously decreasing the longitudinal stability. With this combination for an assumed sinking speed of 25 feet per second, the glide speed would be 112 miles per hour for a wing loading of 30 pounds per square foot.

5. For all combinations, half-span split flaps did not appear promising as a means of increasing lift because of large changes in trim and large unstable deviations in longitudinal stability near the stall.

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6. A variation of Reynolds number from 3,100,000 to 6,900,000 had no appreciable effect on the aerodynamic characteristics of the wing.

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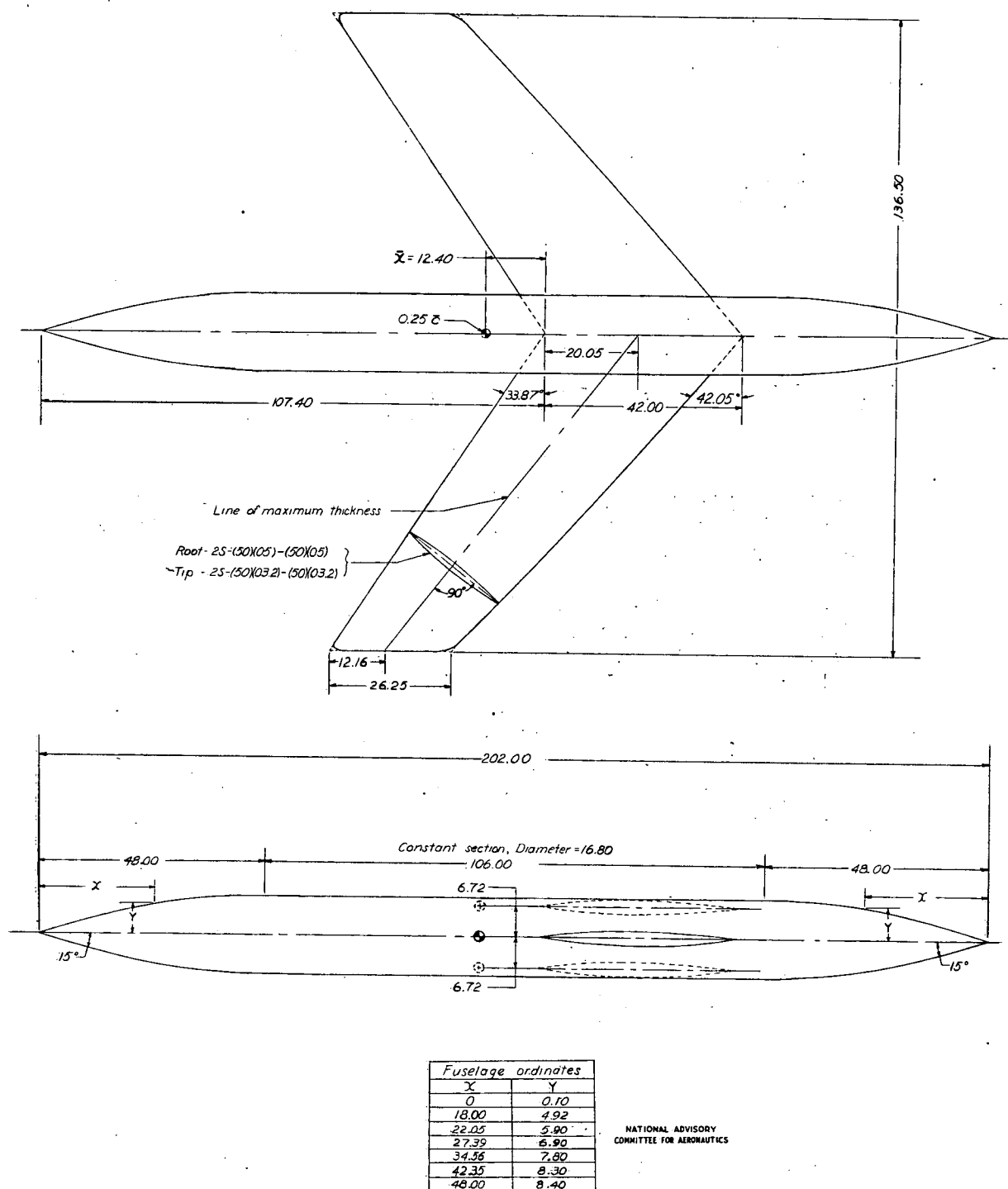


Figure 1.- Sketch of sweptforward wing and fuselage. Wing area, 4728 square inches; mean aerodynamic chord, 35.31 inches; aspect ratio, 3.94. All dimensions in inches.

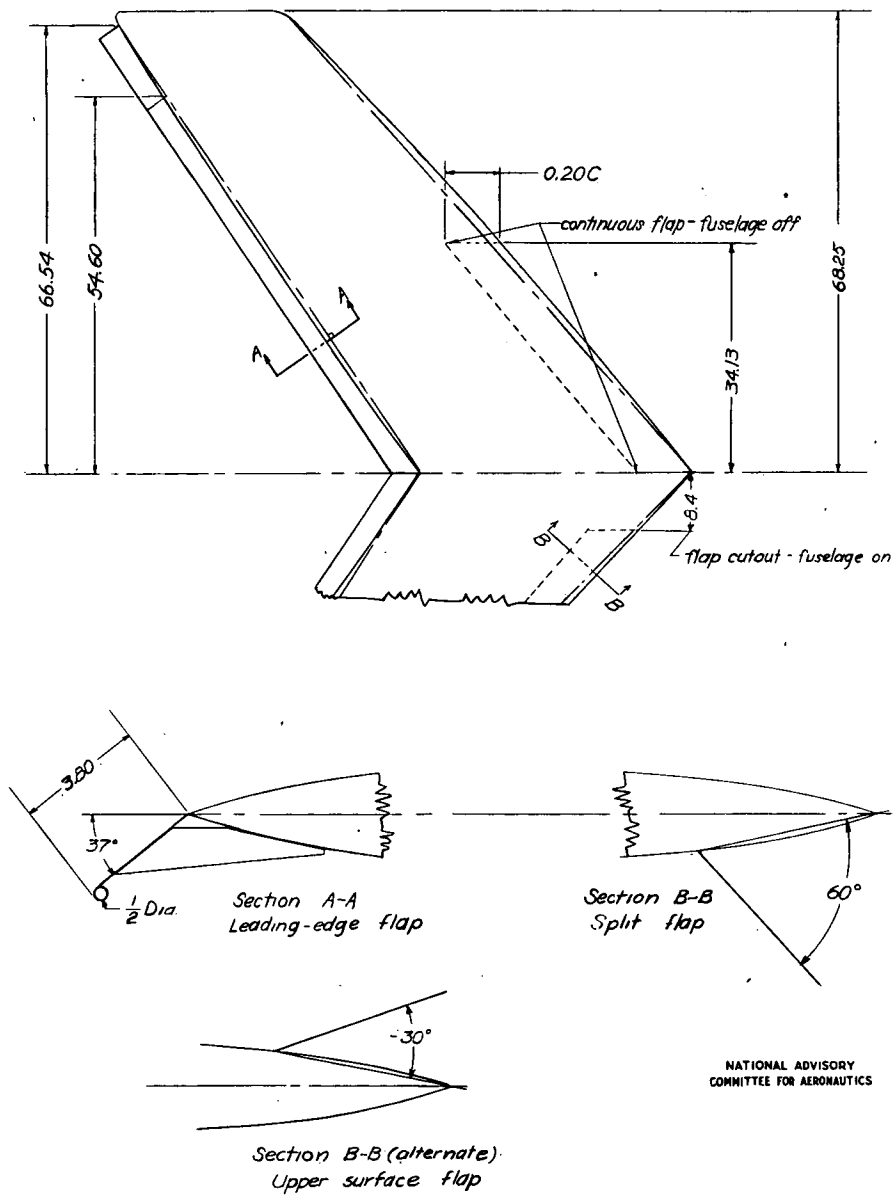
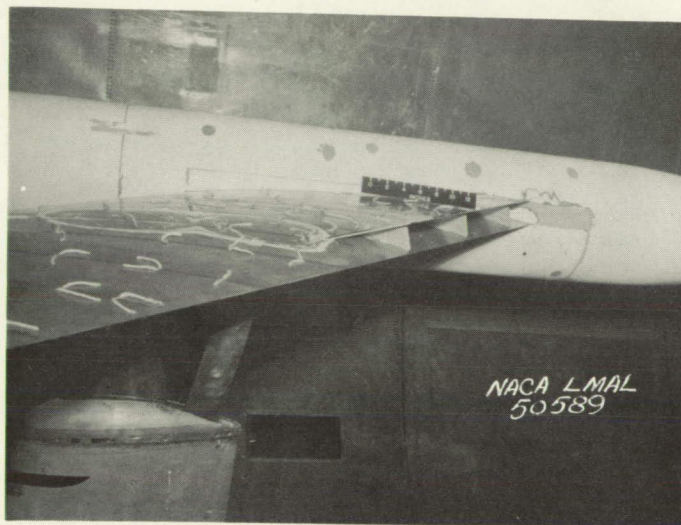


Figure 2.- Installation of the various flaps on the sweptforward wing.
All dimensions in inches.

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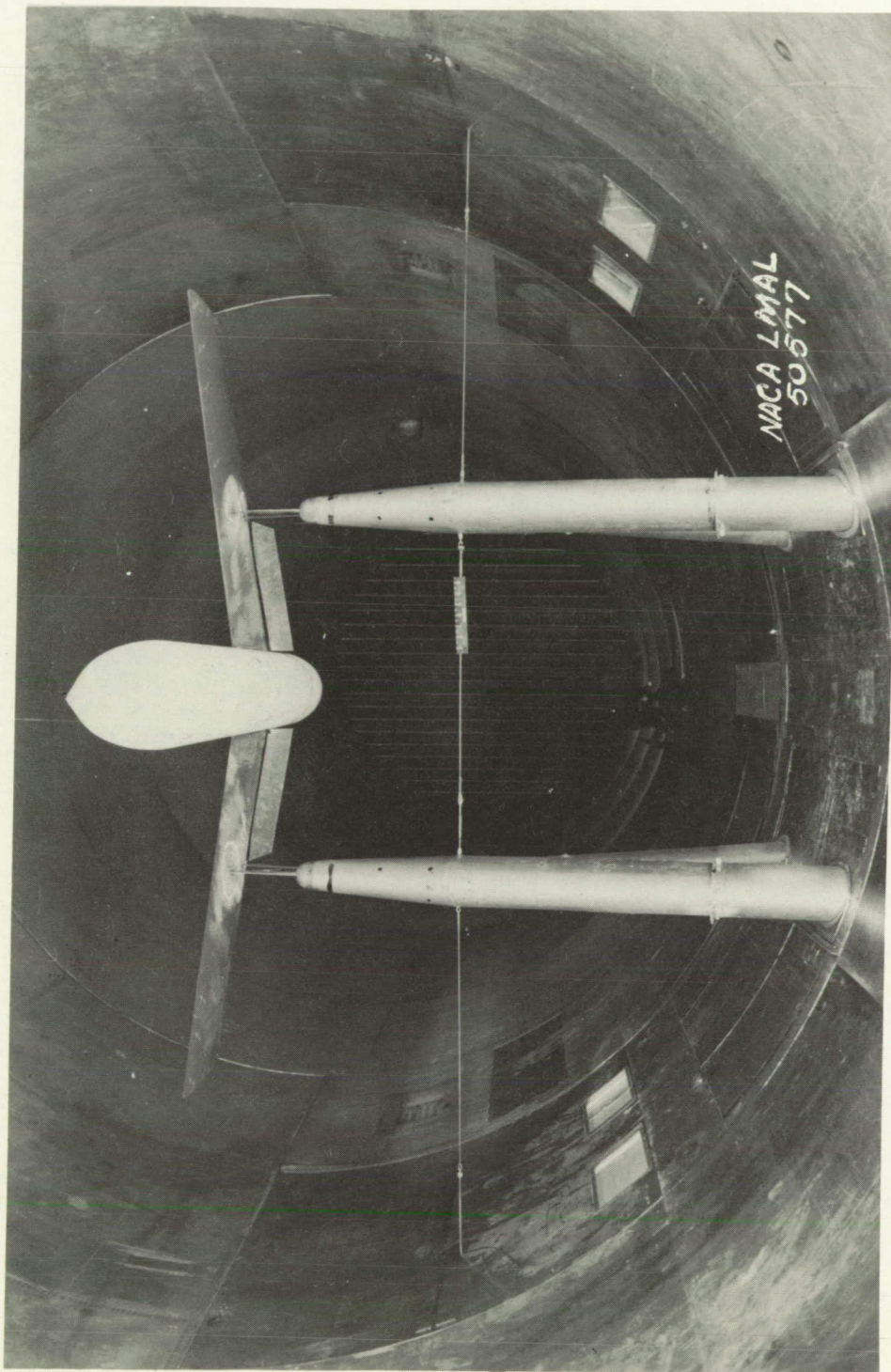


(a) Three-quarter front view of 0.975 $b/2$ span leading-edge flaps.



(b) Side view of upper surface flaps.

Figure 3.- Flap installation on 34° sweptforward wing with midwing fuselage.



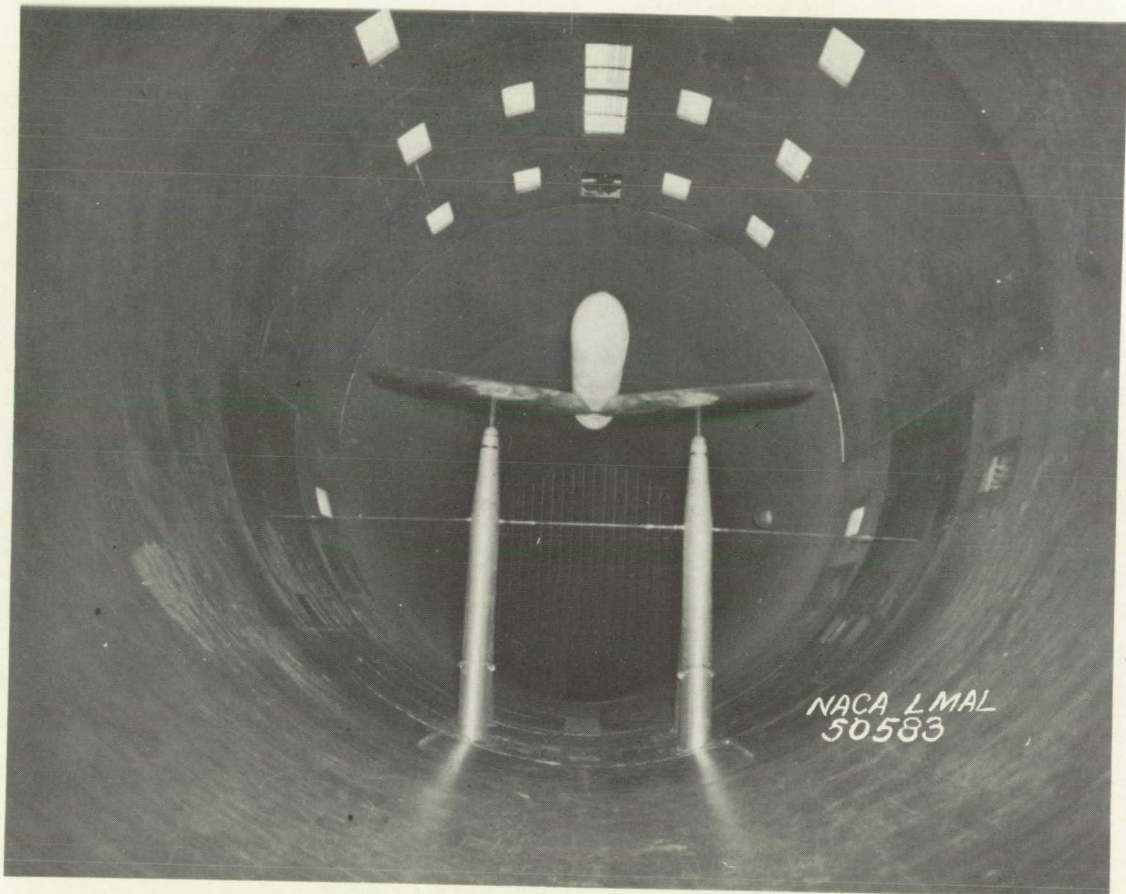
(a) Front view. Midwing-fuselage combination. Split flaps on.

Figure 4.- Sweptforward wing mounted for tests in the Langley 19-foot pressure tunnel.

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(b) Front view. Low-wing fuselage combination.
Flaps off.

Figure 4.- Concluded.

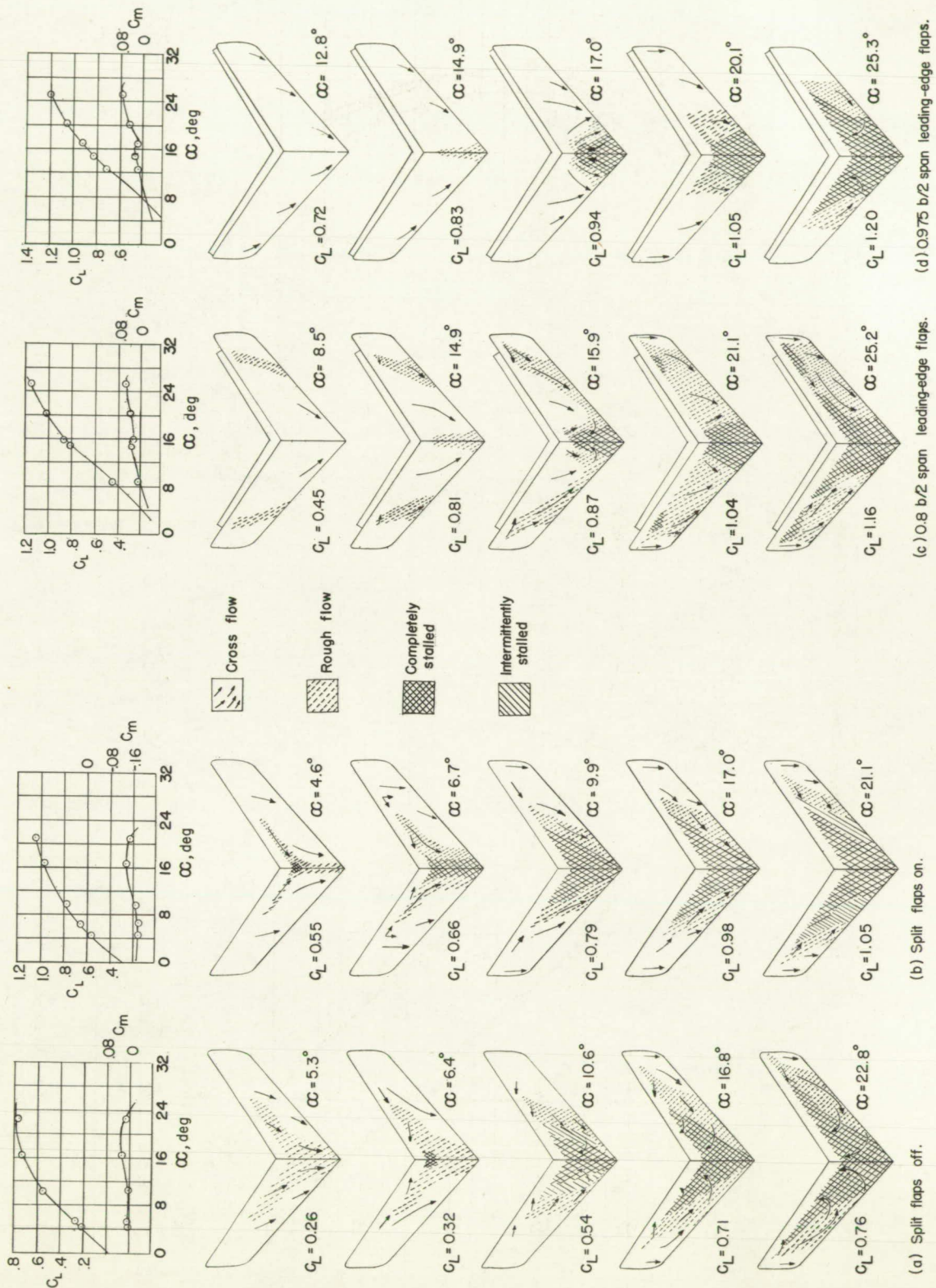
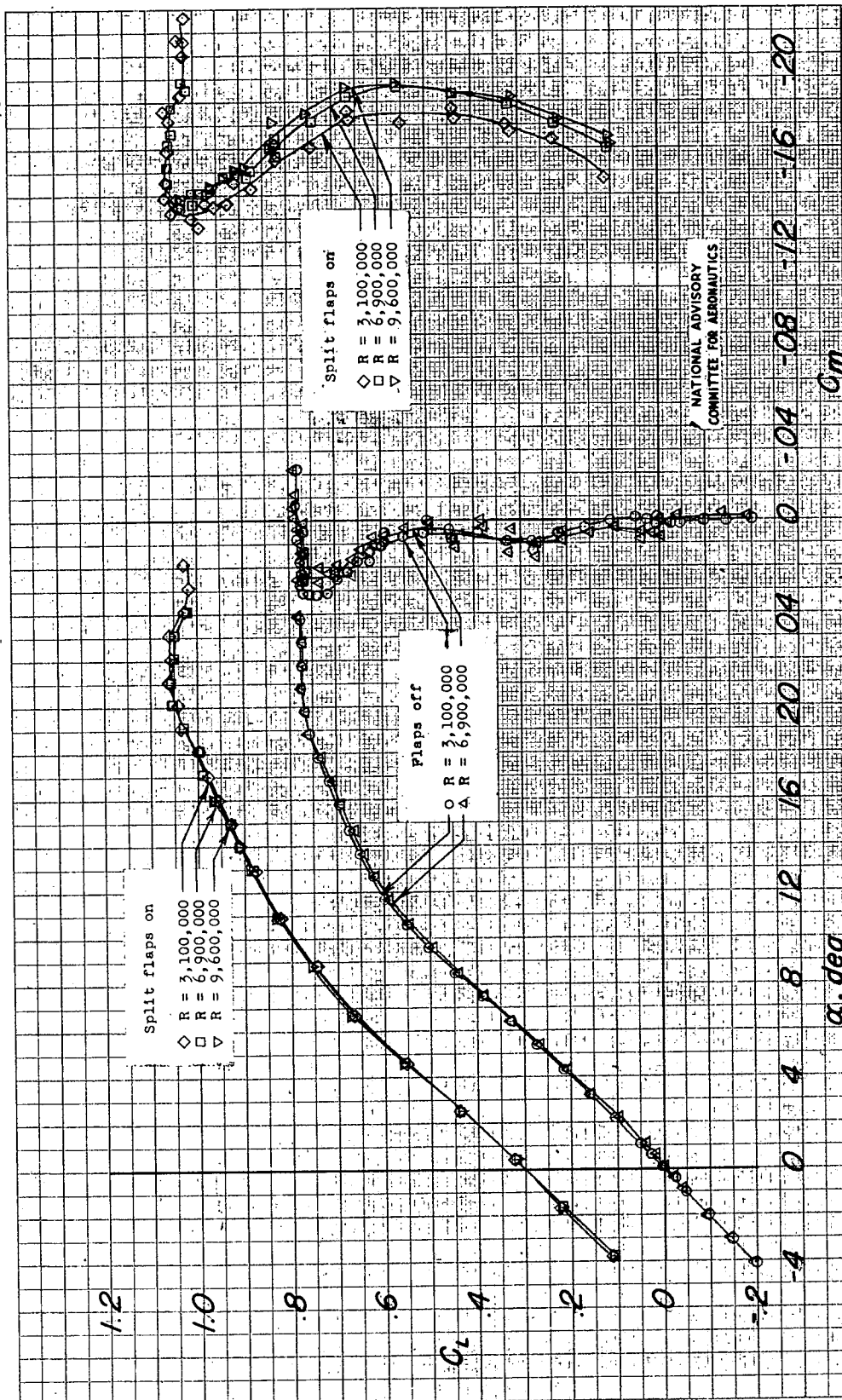
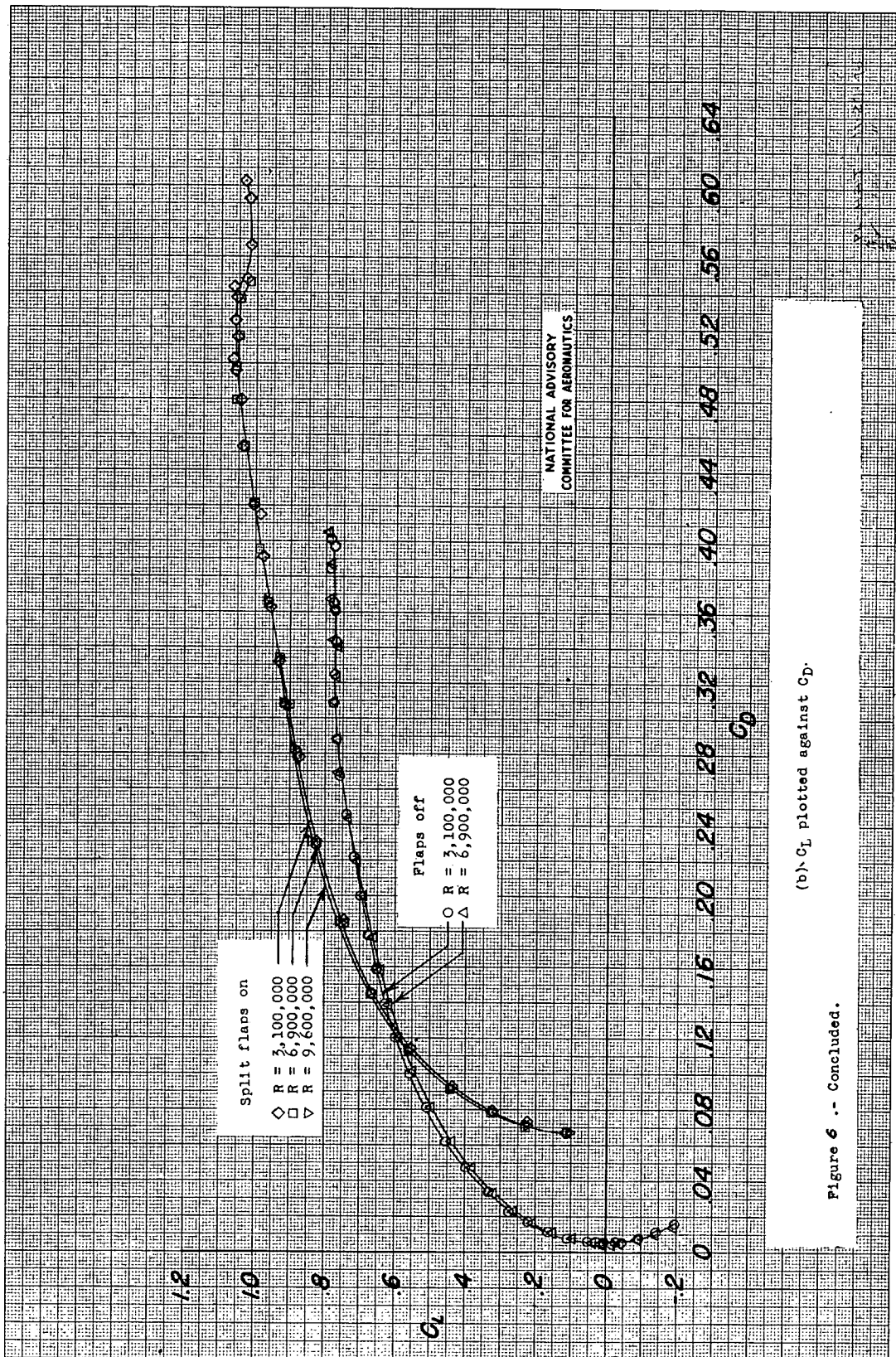
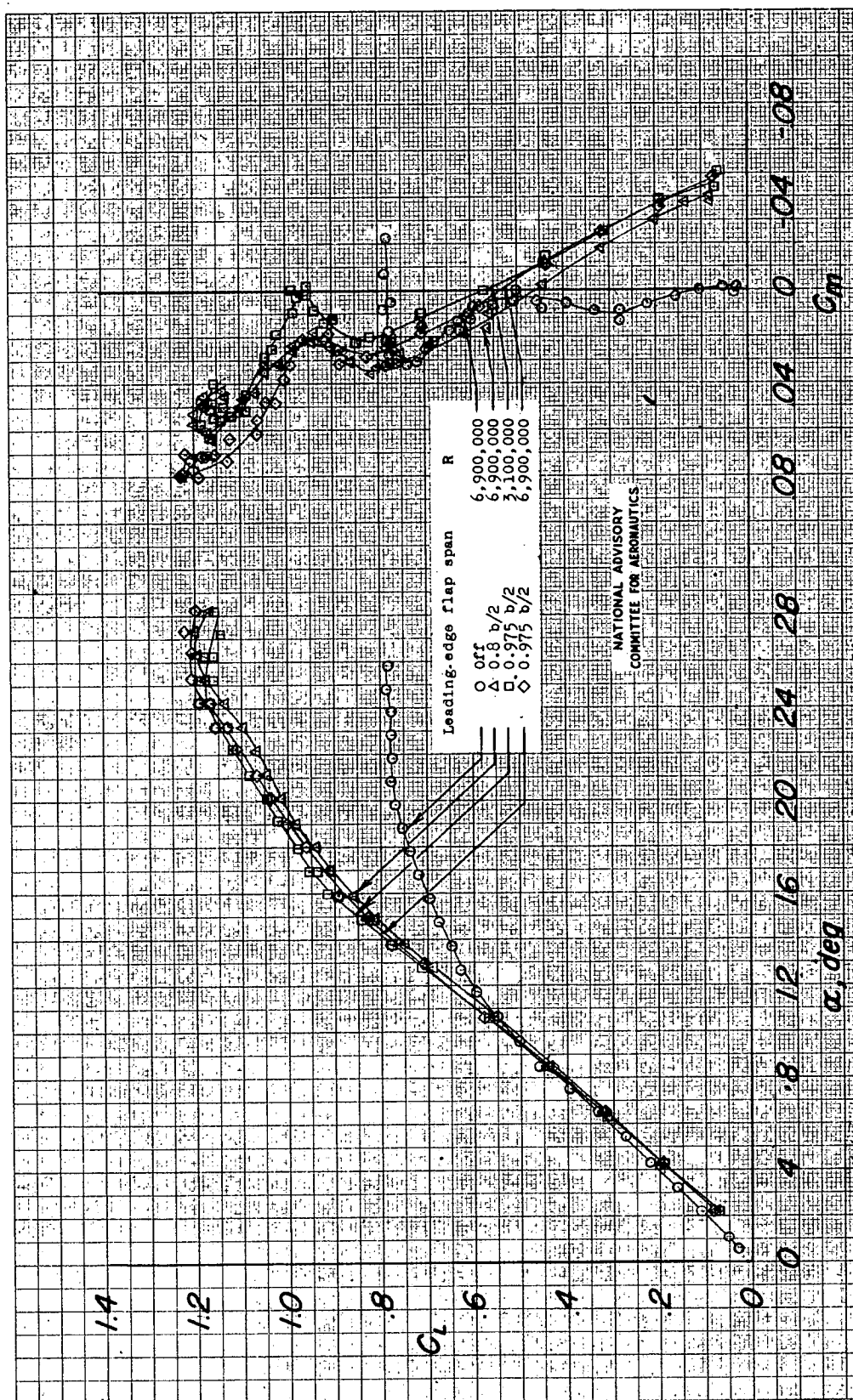


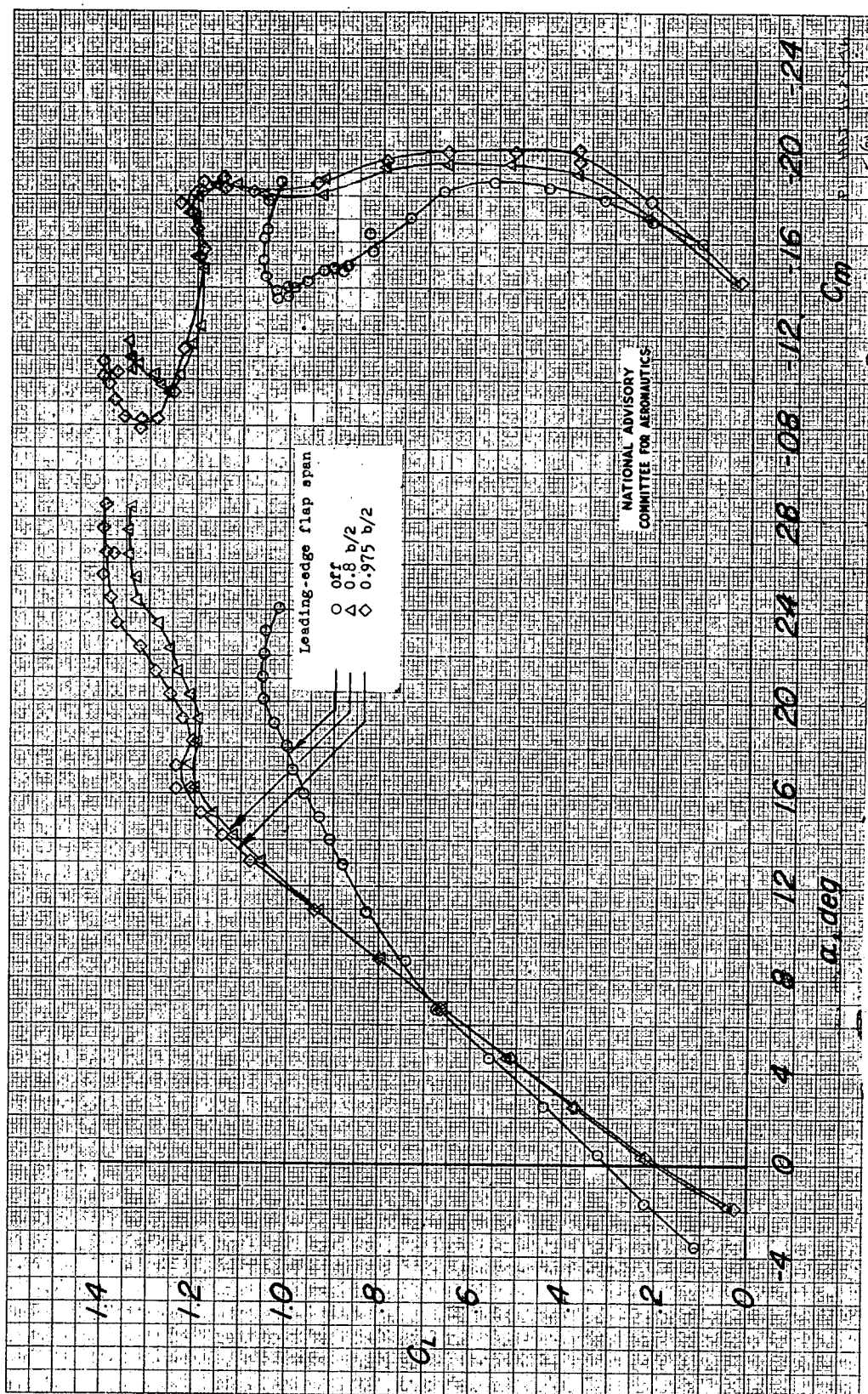
Figure 5 - Stalling characteristics of a sweptforward biconvex wing with various flap arrangements. $R = 6,900,000$.

(a) C_L plotted against α and C_m .Figure 6.- Aerodynamic characteristics of a 34° sweptforward biconvex wing with and without split flaps.



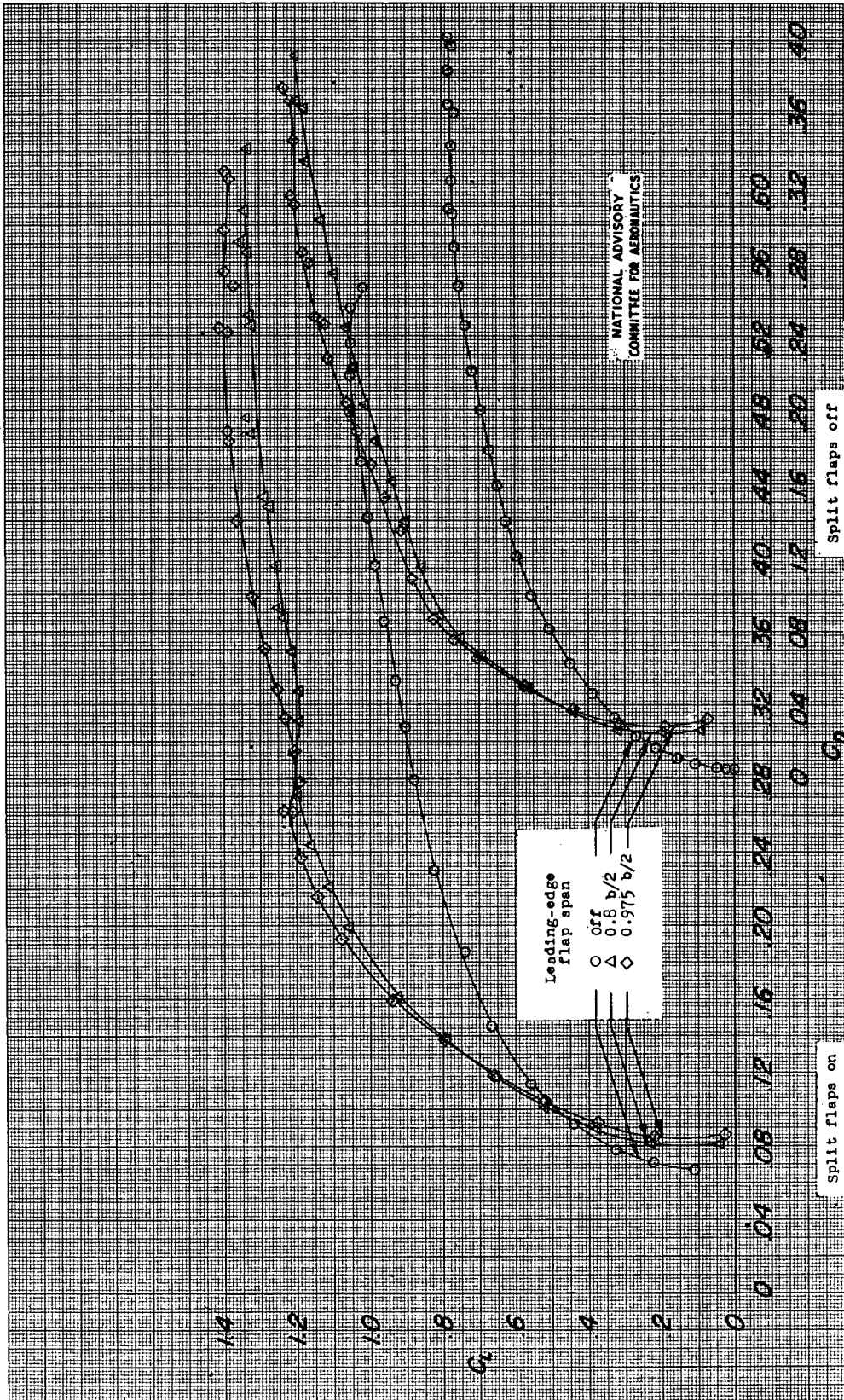
(a) Split flaps off. C_L plotted against α and C_m .

Figures 7.- Effect of leading-edge flaps on the aerodynamic characteristics of a 34° sweptforward bi-convex wing with and without split flaps.



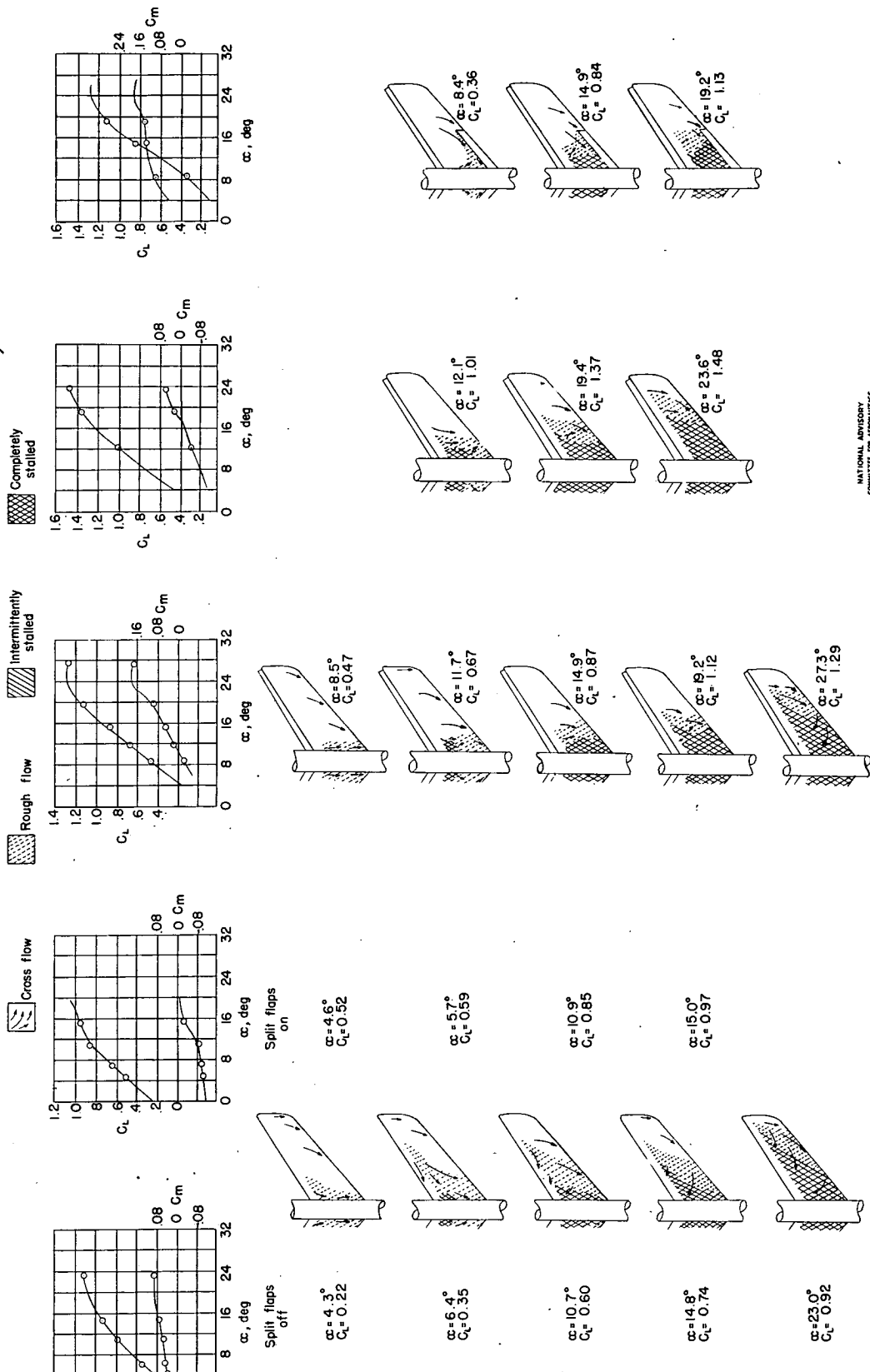
(b) Split flaps on; C_L plotted against α and C_m . $R = 6,900,000$.

Figure 7.- Continued.



(c) C_L plotted against C_D ; $R = 6,900,000$.

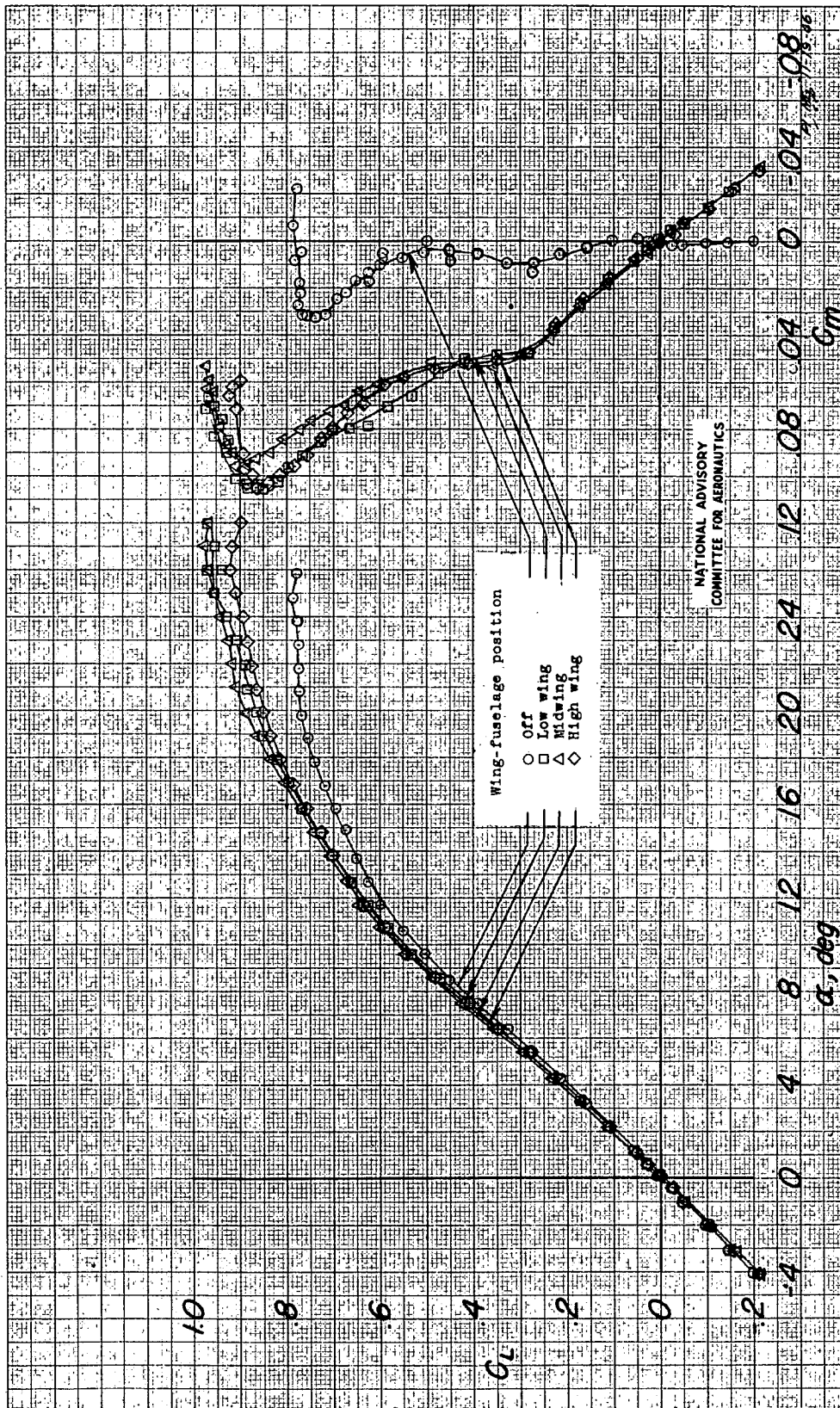
Figure 7.- Concluded.



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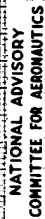
(e) 0.975 b/2 span leading-edge flaps and upper surface flaps.

Figure 8.-Stalling characteristics of a sweptforward biconvex wing with midwing fuselage and various flap arrangements. $R=6,900,000$.



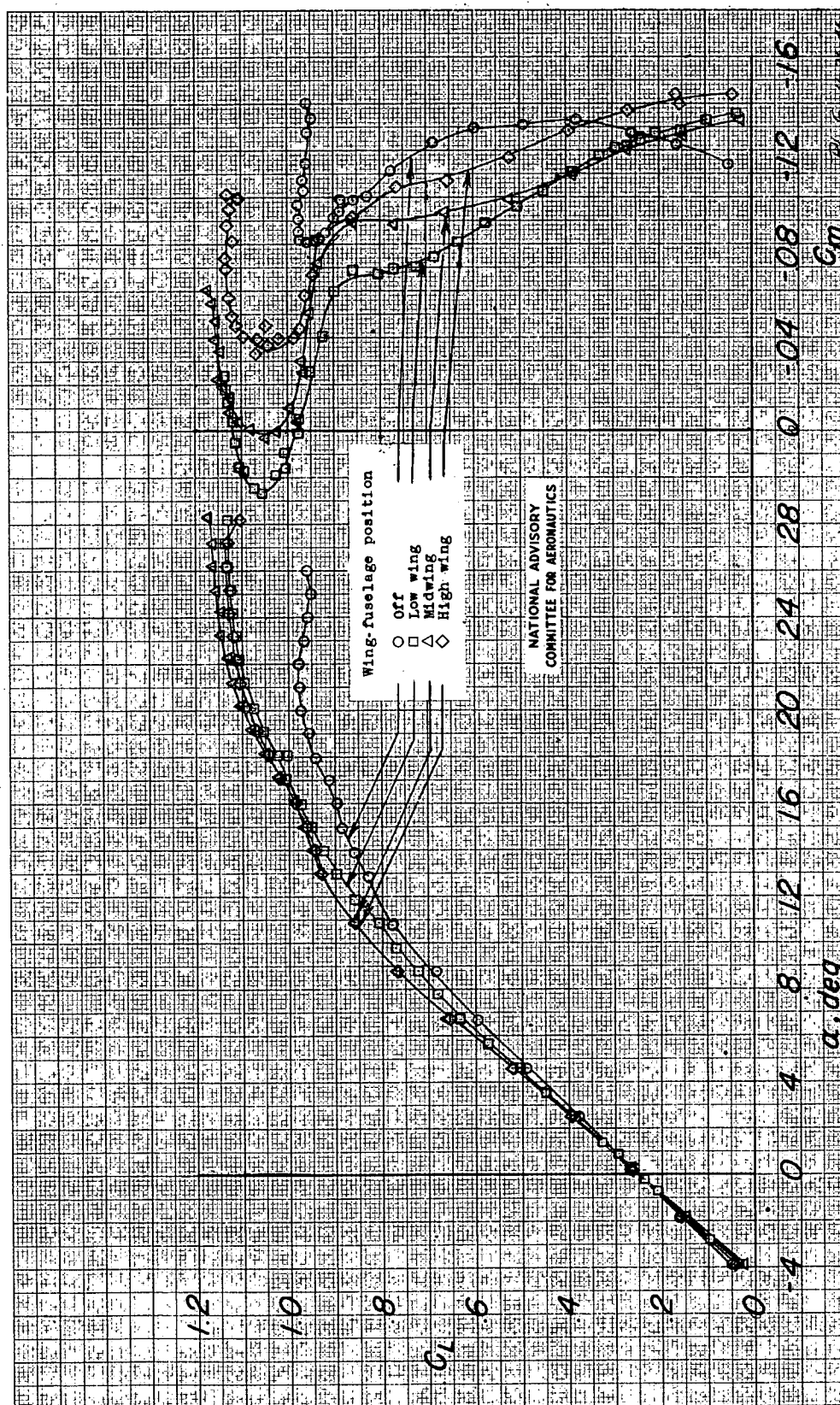
(a) C_L plotted against α and C_m

Figure 9.- Effect of fuselage on the aerodynamic characteristics of a 31° sweptforward biconvex wing. Split flaps off; $R = 6,900,000$.



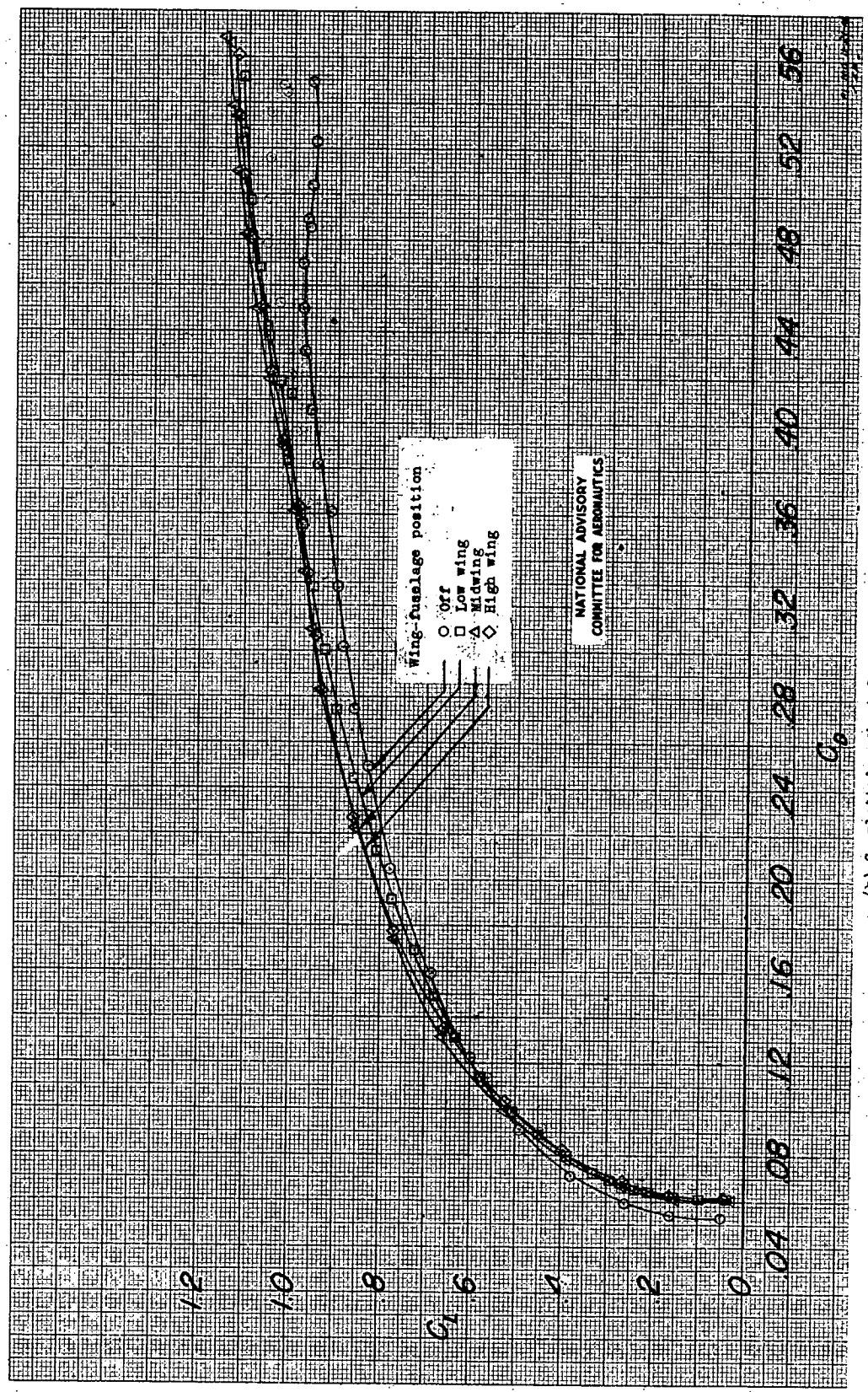
(b) C_L plotted against C_D

Figure 9 .- Concluded.



(a) C_L plotted against α and C_m

Figure 10.- Effect of fuselage on the aerodynamic characteristics of a 34° sweptforward biconvex wing. Split flaps on (center section removed); $R = 6,900,000$.



(b) C_L plotted against C_D

Figure 10.- Concluded.

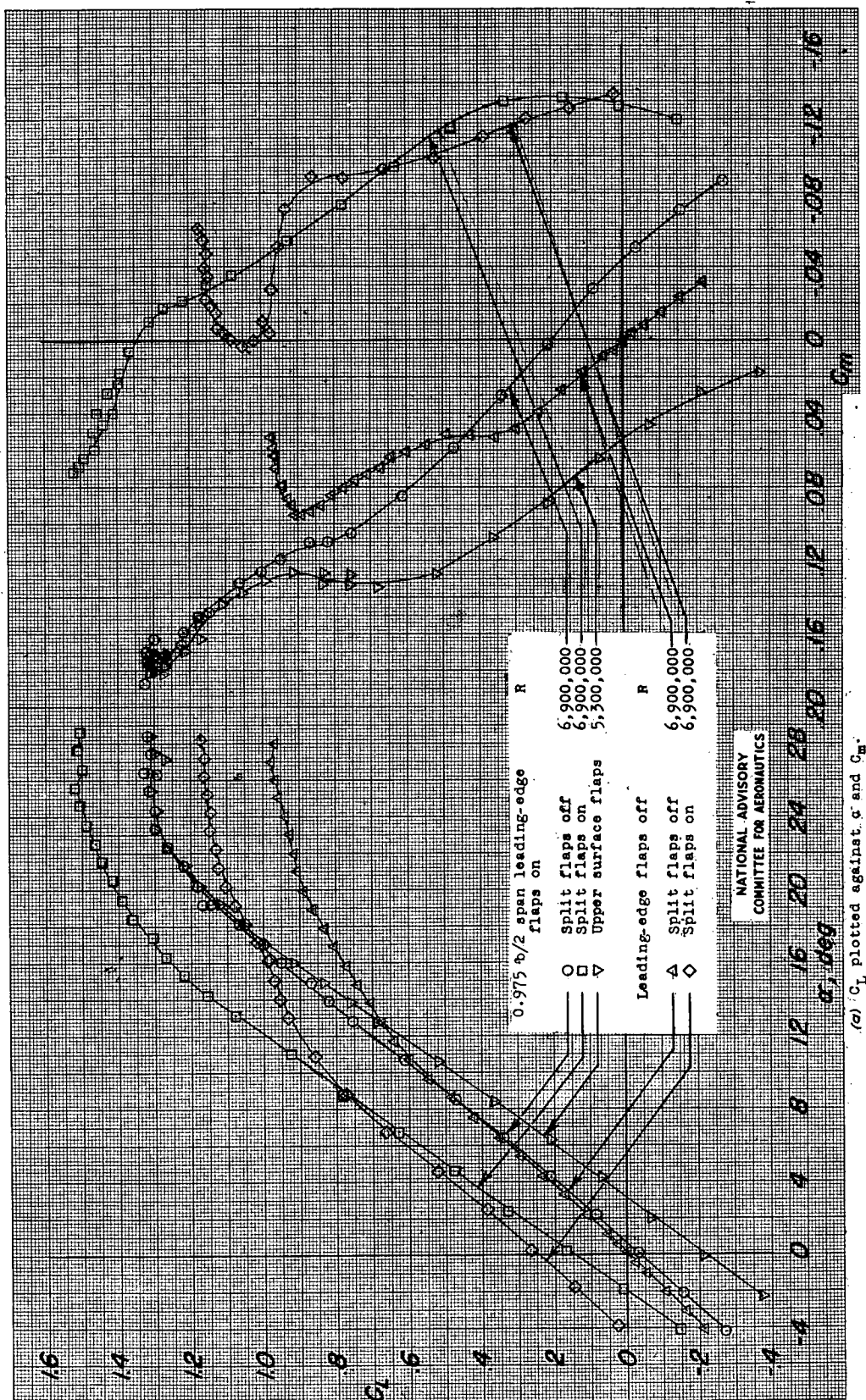
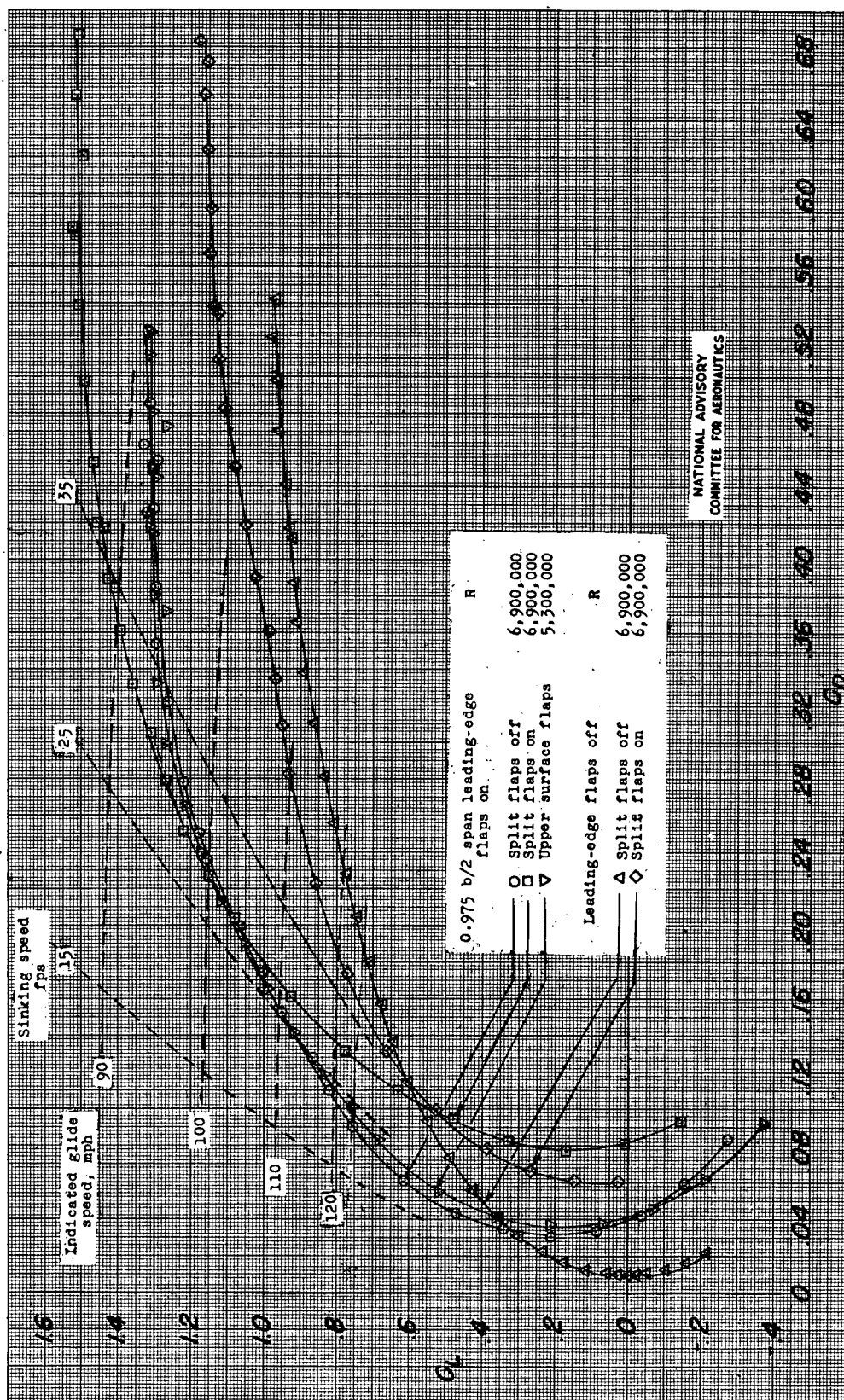
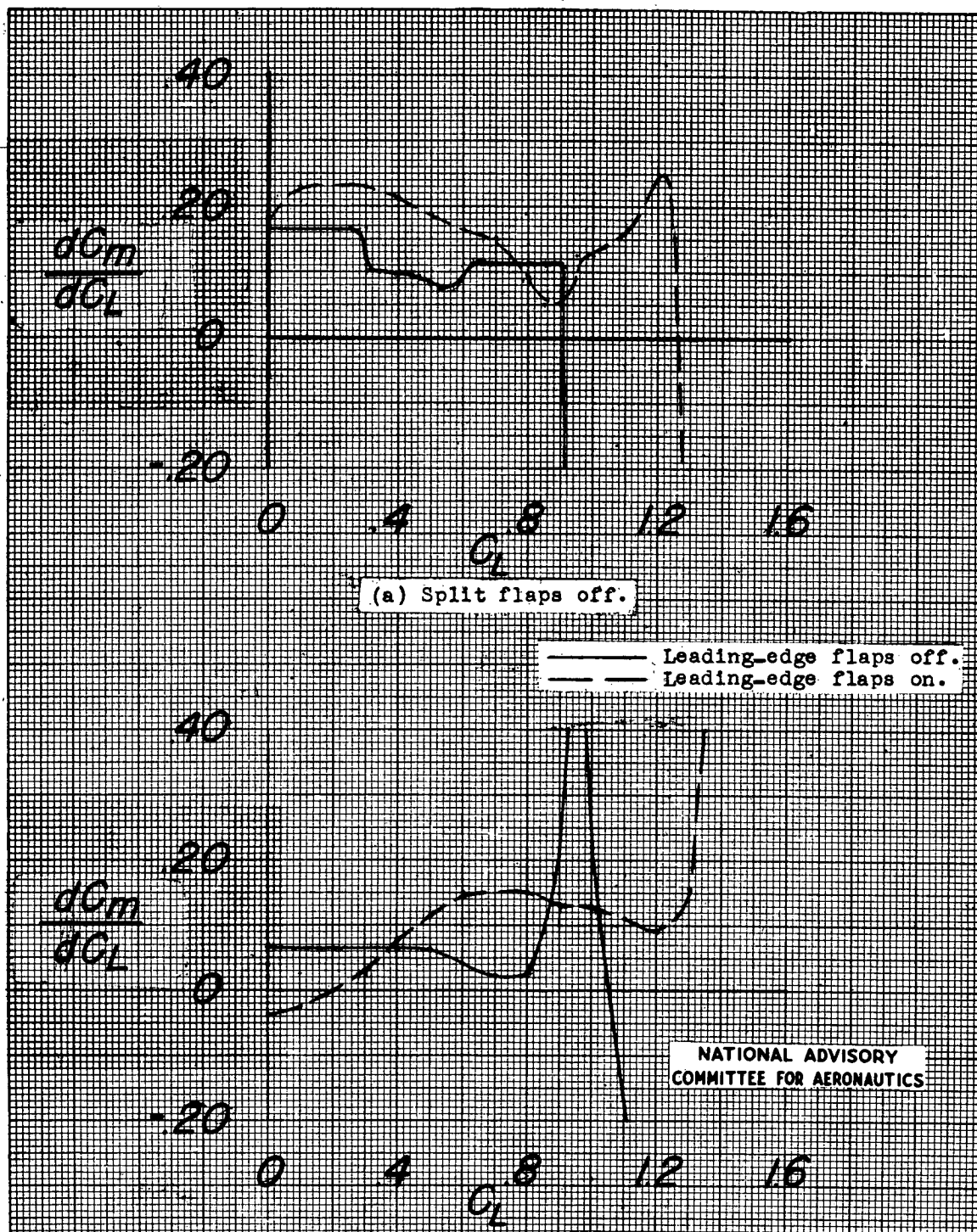


Figure 11.- Aerodynamic characteristics of a 34° sweptforward biconvex wing with midwing fuselage and various flap arrangements (center section of flap removed for split flap configuration).



(b) C_L plotted against C_p .
(With a superimposed grid of glide speed and sinking speed for a wing loading of 30 lb/sq ft)

Figure 11.- Concluded.



(b) Split flaps on.

Figure 12.- Variation of longitudinal-stability parameter $\frac{dC_m}{dC_L}$ with lift coefficient for 34° sweptforward biconvex wing. Midwing-fuselage combination.

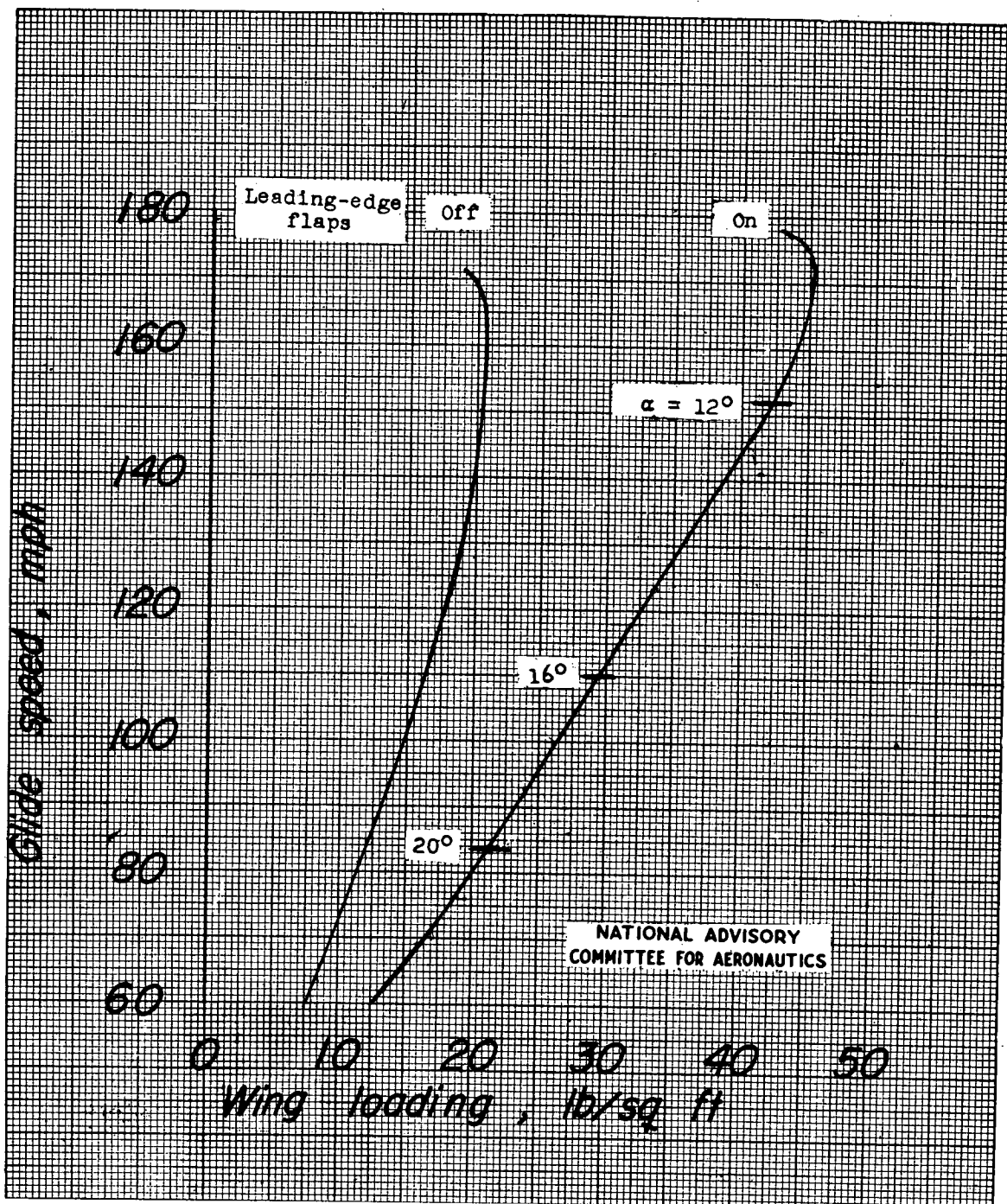


Figure 13.- Variation of glide speed with wing loading at a constant sinking speed of 25 feet per second for a 34° sweptforward biconvex wing with and without full-span leading-edge flaps. Midwing-fuselage combination. Split flaps off.

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